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A computational investigation of the stiffened composite panel with discrete-source damage

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Abstract

The mechanical responses change in the stiffened composite panels because of discrete-source damage. FEA method was used to study the damage propagation and failure properties of stiffened composite panels with discrete-source damage. The influence of discrete-source damage on residual strength of the composite panel was also investigated. It has showed that there is high strain concentration at the notch tip. Based on different failure criteria, FEA method with progressive failure procedure was used to simulate the damage progression and failure procedure of the notched stiffened composite panel effectively. The global models were set up firstly to calculate the stress distribution and evaluate the residual strength of the panels with different cut angles. The local model was set up secondly to simulate the local damage and failure extending both in-plane and interface. The in-plane and interface damage mechanisms could be considered in this model. The research results indicated that notched composite panel was suitable to simulate the damage property of stiffened composite panel with discrete-source damage. The changes in the cut angles are not significant influenced residual strength of the composite panel with discrete-source damage.

Keywords: FEA,discrete-source damage, stiffened composite panel, progressive failure

1. Introduction

Stiffened composite panels are successfully used in primary aircraft structures such as wings and fuselages. The casually caused discontinuous damage such as cutouts and notches changes the mechanical

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responses of stiffened composite panels. Discrete-source damage comes from unknown incidents such that flight maneuvers are limited. It is very important to find out the mechanical responses of stiffened composite panel with discrete source damage. Analyses and tests have been conducted in the composite structures. Numerical methods are used to investigate the damage progression and residual strength of stiffened composite panels with this type of damage in recent years.

Wang[1] demonstrated the progressive failure analysis capability in NASA Langley's COMET-AR finite element analysis code on a large-scale built-up composite structure. Andrew [2] performed finite element analysis, with and without discrete damage, of a composite semi-span test article that represents the Boeing 220-passenger transport aircraft composite semi-span test article. Zhunk [3] analysis the in-plane compression behavior of thin-skin stiffened composite panels with a stress concentrator in the form of an open hole or low velocity impact damage and employed maximum stress criterion to estimate the residual compressive strength of the panel. Brian[4] use of standard path-following non-linear finite element analysis and a Marguerre-type Rayleigh-Ritz energy method to investigate damage tolerant hat-stiffened thin-skinned composite panels with and without a centrally located circular cutout under uniaxial compression loading. Newman [5] and Seshadri [6] used the Structural Analysis of General Shells (STAGS) finite-element shell code and the critical crack-tip-opening angle (CTOA) fracture criterion on stiffened , un-stiffened panels and wide panels residual strength analyses. Adrian [7] conducted numerical investigations into the damage growth and collapse behaviour of four types composite blade-stiffened structures with or without damage. In the numerical analysis of the undamaged panels, collapse was predicted using a ply failure degradation model, and a global-local approach. Gotsis [8] performed computational simulation to S-Glass/epoxy [0/90/±45]S laminated reinforced fibers composite stiffened plate, subjected to thermo-mechanical loads, predicted the damage progression, fracture thought the thickness and propagation to final fracture of the structure. Lauterbach [9] used an analysis tool to investigate fuselage representative carbon fibre-reinforced multi-stiffener panel under compressive loading, predicting interlaminar damage initiation and degradation models for capturing interlaminar damage growth as well as in-plane damage mechanisms. Irisarri [10] presented a multi-objective optimization methodology for composite stiffened panels. The purpose is to improve the performances of an existing design of stiffened composite panels in terms of both its first buckling load and ultimate collapse or failure loads.

The numerical method is conducted to investigate the mechanical responses of stiffened composite panel with discrete source damage. The finite element models were set up to analysis the damage progression and failure procedure of the notched stiffened composite panels. The stress condition of the panel under uniaxial load could be calculated from FEA models. Analytical predictions of a stiffened panel are compared with the test data in order to demonstrate that reasonable test and analysis correlation could be obtained.

2. Panel description

Stiffened composite panels were especially made up of carbon fiber-reinforced epoxy resin laminate. Dimensions of the panel are shown in Fig. 1. The length and width of the panel are 200 and 240 mm, respectively. The height of the stiffeners is 20 mm. and the width of the flanges is 30 mm. This panel has three-stringers and a notch to simulate discrete source damage. The cut was modeled as 80 mm length, 4 mm width and 2 mm radius at tip. The angle α is 30°, 45°, 60°, 75° and 90° in different model. The layer sequences are [-45/0/45/90/0/-45/-45/0/45/0/-45/0/0/45/90/-45/0]s for the skin, $[0_2/90/0/-45/0_4/45/0_4/-45/0_4/45]$ for the stiffener flange and $[0_2/90/0/-45/0_4/45/0_4/-45/0_4/45]$s the stiffener. In the analysis, modulus and strengths for a typical T300/QY8911 unidirectional lamina are used as showed in Table 1.

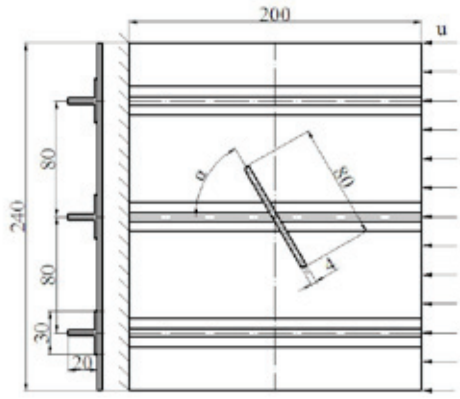

Fig.1 Stiffened composite panel

Table 1 Modulus and strength for lamina

E1/MPa	E2/MPa	ν12	G12/MPa	
124600	7210	0.328	4080	
Xt/MPa	Xc/MPa	Yt/MPa	Yc/MPa	S12/MPa
1239	1081	38.7	189.4	81.2

3. Numerical models and results

The finite element models were created with the commercial code ABAQUS. The geometry dimensions are according to the panel shown in Fig.1. Fix boundary conditions were applied at the left edge, free boundary conditions were applied at the top and under edge, and a uniform longitudinal displacement load was applied at the right edge of the finite element model. Note that loads were applied both on the skin bays and the blades to simulate the actual loading conditions. The finite element mesh is much more refined in the local notch region to allow for an accurate representation of the stress field in the vicinity of the notch tip. The shell element is used in the global model analysis models and solid element is used in the local model analysis.

3.1. Global models analysis

FEA models were set up and with different notch angle which are 30°, 45°, 60°, 75° and 90°. All the models were conducted using the material properties data in Table 1. The stress distribution was calculated from the models. Take 45° model for example, the mises stress contours obtain from the analysis at an applied load of 100kN are showed in Fig.2. The maximum stress values and the average stress values in 0 ply of skin are listed in the Table 2. The stress concentration is apparent in the vicinity of the notch tip in the notched panels. The mechanics responses changed in the panel because of discrete source damage.

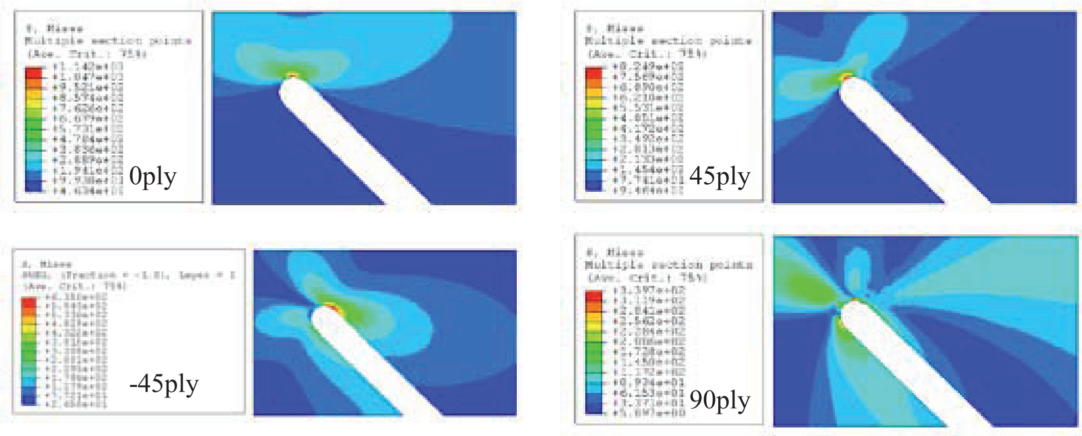

Fig.2 stress contours in local area (45° model)

Table 2 analysis results

α criterion	30°	45°	60°	75°	90°
maximum stress value/MPa	1015	1142	1085	997.7	1090
average stress value/MPa	126.7	131.5	134.1	134.5	135.8
ultimate load /kN	458.5	433.7	412.8	406.1	406.5

The damage would not happen in the model until the applied load condition increased into certain value. An iterative approach was used in the ABAQUS procedure for material nonlinear analysis. The material properties of every element has different stiffness status which controlled by a user-written subroutine. The damage initials from the vicinity of the notch tip and extends along the weakest section. For each load step, the stress values were extracted through a user subroutine USDFLD contained failure criteria and material strength properties which are showed in Table 1. The degraded material properties can be classified into three types according to the failure mechanisms. The three basic failure mechanisms are matrix cracking, fiber damage and fiber-matrix shearing. The failure analysis must be able to predict the failure mode in each ply, and then apply the corresponding reduction in material stiffness as the loading level is increased. Hashin criterion is used here in the prediction. Based on the classical lamination theory and progressive failure analysis, the behavior of laminates under plane stress loading is predicted. The results are presented in Table 2.

3.2. Local models analysis

The solid elements were used in the local analysis. Firstly the solid global model was set up to figure out the boundary condition of the local area. Secondly the local model, which was cut from the solid global model, had separate element layer and cohesive elements between the plies. The progressive failure analysis combined with cohesive damage analysis in the solid local model gives the details of the damage mechanism in qualitative. The damage initials from the vicinity of the notch tip and the ply element stiffness degraded, simultaneously the interlaminar cohesive element damage initialed and propagated on the energy-based damage evolution law. The procedure of the local model is shown in Fig.3.

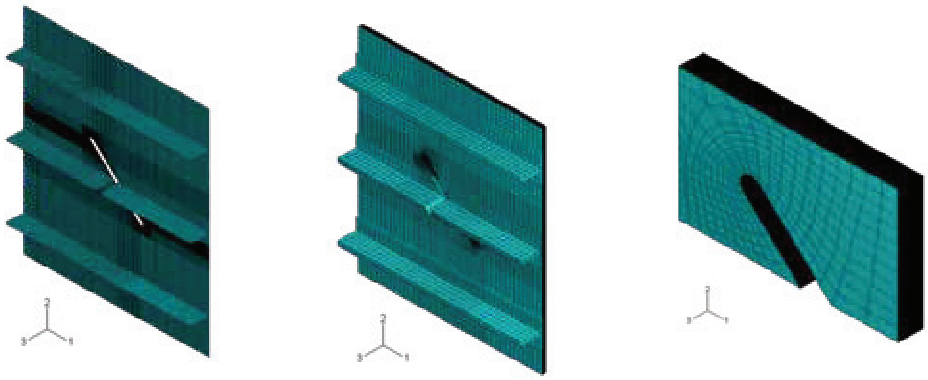

Fig.3 finite element model

4. Conclusions

The numerical models were built to simulate the damage progression and failure procedure of the stiffened composite panels with discrete-source damage. The global linear models associated with local nonlinear models were set up to predict the mechanics responses but also the damage mechanism. Taken the failure criteria into the models, the strengths of the panels with different notch were predicted from the simulation. As a result, the conclusions are showed below.

(1) The discrete-source damage significantly influences the stiffness and the strength of the stiffened composite panels. The computation results show obviously concentration at the tip of the notch.

(2) The angle of the notch affects the residual intensity because the section area change, but indistinctively in respect that stiffener reduces the sensibility to the skin cut.

(3) The global model can be used in the strength prediction of the stiffened composite panels with discrete-source damage, and the local model can be used to predict the failure modes.

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